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TECHNICAL MEMORANDUM

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EFFECTS OF BASE BLEED FLOW ON BASE REGION TEMPERATURES AND PRESSURES OF SEVERAL SIMULATED MISSILE AFTERBODY CONFIGURATIONS - MACH NUMBER RANGE OF 0.8 TO 2.0

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Cleveland, Ohio

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EFFECTS OF BASE BLEED FLOW ON BASE REGION TEMPERATURES AND PRESSURES OF SEVERAL SIMULATED MISSILE AFTERBODY CONFIGURATIONS - MACH NUMBER RANGE OF 0.8 TO 2.0*

By Alfred S. Valerino, Arthur M. Shinn, Jr., and Bruce G. Chiccine

SUMMARY

An investigation was conducted through a Mach number range of 0.8 to 2.0 to determine the effects of base bleed flow on the base region temperatures and pressures of several simulated missile afterbody configurations of cylindrical cross section. Nominal afterbody diameter was approximately 8 inches. A liquid-oxygen - JP-4 fuel rocket engine operating at a chamber pressure of 500 pounds per square inch was employed to obtain a hot exhaust jet. Configurations having engine extensions of 0.32 and 0.58 caliber downstream of the afterbody base and an engine retraction of 0.20 caliber upstream of the base were investigated. Base bleed flow was obtained from a source external to the tunnel or through afterbody slots or scoops. Configurations utilizing scoops also employed a short turbine exhaust duct projecting through the base, through which propane, simulating fuel-rich turbine exhaust products, was discharged into the base region. The turbine exhaust configuration was also tested without bleed flow.

Results of the investigation indicated that, with the 0.32- and 0.20-caliber afterbody configurations, the use of base bleed flow decreased base region temperatures from values as high as 800° and 2000° F, respectively, to values on the order of ambient temperature, approximately 150° F. Temperatures of the turbine exhaust configuration operating at a Mach number of 0.8 were reduced from 1950° to approximately 250° F by utilizing bleed flow.

The engine external pressure distributions were affected by the manner in which bleed flow was ducted into the engine compartment. For example, large pressure gradients were induced on the engine of the 0.32-caliber afterbody configuration by the use of flush afterbody slots or scoops 0.5 inch in height. Such gradients were reduced considerably or eliminated with the use of scoops 0.2 inch in height or with metered bleed flow obtained from a source external to the tunnel. Metered bleed flow was also effective in reducing the engine aerodynamic moments of the 0.58-caliber afterbody configuration that were caused by the external stream flow.

*Title, Unclassified.

INTRODUCTION

The combination of a hot rocket jet containing unburned fuel and the blunt base of a ballistic missile provides an environment for which base region temperatures high enough to damage equipment housed in the engine compartment are possible (ref. 1). One method of protecting the engine compartment is to install a firewall. Such a heat barrier, however, adds weight to the missile, which reduces the final velocity or range. Other disadvantages include the mechanical complications of installation due to introducing a seal around a gimbaled engine and the possibility of an explosion in the engine compartment due to leaks in the propellant lines. High base region temperatures can also be alleviated by increasing the rocket-engine projection from the afterbody base as discussed in reference 1.

This report will discuss still another approach, namely, the use of bleed airflow through the engine compartment. Base region temperatures and rocket-engine external pressure distributions of several simulated missile afterbody configurations were investigated in the Lewis 8- by 6-Foot Supersonic Wind Tunnel. Bleed airflow through the engine compartment was obtained either from a source external to the tunnel or by the use of afterbody slots or scoops.

Test data were obtained through a Mach number range of 0.8 to 2.0 at angles of attack of 0°, 2.5°, and 5° and at rocket gimbal angles of 0° and 4°. The weight-flow ratio of liquid oxygen to JP-4 fuel of the rocket engine was varied from approximately 1.5 to 2.6 at a combustion-chamber pressure of approximately 500 pounds per square inch.

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SYMBOLS

A_e	engine nozzle-exit area, 0.063 sq ft
C_m	engine aerodynamic moment coefficient, $m/qA_e l$
D	diameter
l	distance from engine gimbal point to nozzle-exit plane, 7.25 in.
M	Mach number
m	engine aerodynamic moment about gimbal point due to engine external pressures
o/f	ratio by weight of liquid oxygen to fuel
P	total pressure

p static pressure

q dynamic pressure

r radius

T total temperature, $^{\circ}F$

w weight flow, lb/sec

w_0 weight flow in free-stream tube having an area equal to open area (0.1675 sq ft) at afterbody base

x distance upstream from plane of rocket-engine exit

α model angle of attack, deg

β engine gimbal angle in pitch plane to correct for angle of attack, deg

Subscripts:

b base of model

bb base bleed

e rocket exit

r external rocket-engine surface

t throat

0 free stream

APPARATUS

The investigation was conducted in the transonic test section of the NASA Lewis 8- by 6-Foot Supersonic Wind Tunnel. The model was attached to the tunnel walls by 45° forward swept struts. A schematic diagram and photograph of the model assembly are shown in figures 1 and 2.

The hot exhaust jet was obtained by employing a water-cooled rocket engine with a bellmouth-shaped nozzle having an exit- to throat-area ratio of 8. The contour and coordinates of the rocket-engine exhaust nozzle are shown in figure 1(b). Rocket-engine combustion-chamber temperature, exit pressures, and characteristic velocities as well as details on the firing of the rocket engine are discussed in reference 1.

Three basic afterbody configurations were studied. They are designated herein as the 0.32, the 0.58, and the -0.20 afterbody configurations; the numbers refer to the extension or retraction, in model base diameters, of the exhaust-nozzle exit from the base plane of the afterbody.

0.32 Afterbody Configurations

The 0.32 afterbody configuration (fig. 3) had a base diameter of 7.87 inches and a rocket-engine projection of 2.53 inches downstream of the afterbody base plane. Bleed flow was obtained from two different sources: the laboratory service air system external to the tunnel and through afterbody slots or scoops. The ratio of the slot area to the open area at the base plane of the slotted afterbody (fig. 3(a)) was 0.16. Two scoop heights of 0.2 and 0.5 inch (figs. 3(b) and (c)) having ratios of scoop frontal area to open area at the afterbody base of 0.007 and 0.017, respectively, were also alternately employed in conjunction with the slots to provide bleed flow. The 0.2-inch-high scoops were obtained by bending the inlet portion of the larger scoops inward to a height of 0.2 inch. Baffle plates, projecting from the scoops into the engine compartment, were utilized to diffuse the bleed flow entering the engine compartment. When the bleed flow was obtained from a source external to the tunnel, it was ducted through the hollow model support struts and then through the engine compartment. This method was used primarily to simulate a well-designed bleed system. Temperatures of the bleed air from the external source, measured in the model near the struts, were of the order of 100° F.

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A turbine exhaust duct, which was simulated with a $\frac{5}{8}$-inch-diameter tube discharging propane into the base region at a total pressure of 23 pounds per square inch and at a temperature of approximately 100° F, was employed with the afterbodies utilizing scoops. The turbine exhaust duct was also tested with the 0.32 afterbody configuration (fig. 3(d)) which did not utilize base bleed flow. With this configuration, a firewall was employed to protect the engine compartment. Photographs of the 0.32 afterbody configurations tested are presented in figure 4.

Air temperatures inside the engine compartment, near the afterbody base plane and at a rake station 0.4 inch downstream of the base plane, were recorded by thermocouples shown in figure 3(d). Rocket-engine external pressure distributions were obtained from the two rows of static-pressure orifices also shown in figure 3(d). Afterbody base pressures were determined from the average of four base static-pressure orifices.

Base bleed flows, passing through the scoops mounted on the afterbody, were estimated from boundary-layer profiles (fig. 5) and by the procedure discussed in reference 2. The bleed flows obtained from an external source were metered by ASME sharp-edged orifices. All weight flows are referenced to a free-stream tube having an area equivalent to the open area at the afterbody base.

0.58 Afterbody Configuration

A schematic diagram of the 0.58 afterbody configuration tested is shown in figure 6. The afterbody base was 8.12 inches in diameter and was located 4.75 inches upstream of the exhaust-nozzle-exit plane. Bleed air was obtained from a source external to the tunnel and ducted through the engine compartment. This configuration was tested at a gimbal angle of 4° only.

As discussed in reference 1, the base heating problem did not exist with the 0.58 afterbody configuration. Therefore, no temperature data will be presented.

Two rows of 12 static orifices located externally on the leeward and windward sides of the engine were employed to determine axial engine external pressure distributions and engine aerodynamic moments. The moments were computed by assuming a linear circumferential pressure distribution between the windward and leeward sides of the engine and integrating the resultant local pressure forces circumferentially and axially along the external surface. The moments are presented herein in coefficient form C_m.

-0.20 Afterbody Configurations

This configuration simulates the afterbody of a missile which utilizes a jet vane control mechanism. The base of the afterbody therefore extends downstream of the exhaust-nozzle exit. Schematic diagrams and photographs of the -0.20 afterbody configurations are presented in figures 7 and 8, respectively. The base had a diameter of 7.04 inches and extended 1.42 inches downstream of the engine exit plane (fig. 7(a)). To simulate the support structure for the jet vane control mechanism, a ring of solid cast iron was attached internally in the base region as shown in the photograph of figure 8(a). Cast iron was selected in view of its heat sink capacity and thus its ability to protect the configuration from being damaged by the high-temperature exhaust gases.

To provide base bleed flow, a slotted afterbody (fig. 7(b)) was employed, and the solid-cast-iron ring was replaced by three 30° cast-iron sections as shown in the photograph of figure 8(b). The ratio of the slot area to the open area at the engine exit plane was 0.784.

Air temperatures of the slotted and nonslotted afterbodies were measured at the base and along the surface of the diverging portion of the cast-iron ring or sections. In addition, with the nonslotted afterbody, air temperatures were measured by thermocouples located 1/8 inch downstream of the divergent portion and at the base of the cast-iron ring as shown in figure 7(a). With the slotted afterbody, additional temperature data were obtained by means of a rake located 0.96 inch upstream of the engine exit plane (fig. 7(b)).

Base pressures were determined from an average of three base static-pressure orifices.

PROCEDURE

The investigation was conducted over a Mach number range of 0.8 to 2.0 through a 0°- to 5°-angle-of-attack range and at pressure altitudes which ranged from approximately 9000 feet at Mach 0.8 to 37,000 at Mach 2.0. The rocket engine was operated at a combustion-chamber pressure of 500 pounds per square inch through a range of liquid-oxygen to fuel (JP-4) weight ratios of 1.5 to 2.6. Firing time of the rocket engine per data point was about 70 seconds; approximately 45 to 50 seconds were required for thermocouples to reach apparent equilibrium temperatures, and the remaining time was needed for recording steady-state pressure data. The engine gimbal angles tested were 0° and 4°. The configurations tested and the test variables are shown in the following table:

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Configuration	Source of bleed flow	Mach number, M_0	Angle of attack, α, deg	Engine gimbal angle, β, deg
0.32 afterbody	Slots	2.0 1.6 .8	0	0
0.32 afterbody with turbine exhaust	0.5-inch-high scoops	2.0 1.6 1.3 .8	0 5	0
0.32 afterbody with turbine exhaust	0.2-inch-high scoops	2.0 .8	0 5	0
0.32 afterbody with turbine exhaust	None	0.8	0 5	0
0.32 afterbody	Metered flow from external source	2.0 1.4 .8	0 2.5 5	4
0.58 afterbody	Metered flow from external source	2.0 1.4 .8	0	4
-0.20 afterbody	None	2.0 1.4 .8	0 5	0
-0.20 afterbody	Slots	2.0 1.4 .8	0 5	0

DISCUSSION

Temperatures

The temperatures presented in this report are the maximum of those measured at the indicated locations of the configurations. These maximum temperatures were not always indicated by the same thermocouple for each rocket firing. It should be noted that the reported temperatures may not agree exactly with those which are experienced in flight since the stream total temperatures of this study differ somewhat from flight values at the corresponding Mach numbers and altitudes as indicated in the following table:

Mach number, M_0	Altitude, ft	Total temperature, $^{\circ}F$	
		Tunnel nominal	Flight NASA standard
2.0	37,000	165	246
1.6	29,000	140	169
1.4	25,000	130	139
1.3	22,500	128	132
.8	9,000	110	89

<u>0.32 Slotted afterbody</u>. - A comparison of the base region temperatures of the slotted afterbody configuration and the configuration having a closed base to simulate a firewall (ref. 1) at an engine gimbal angle of 0° is made in figure 9. At Mach numbers of 2.0 and 1.6, the base region temperatures with base bleed were approximately equal to free-stream total temperatures (130° to 160° F), a temperature approximately 450° F less than those obtained with the closed-base configuration. At a Mach number of 0.8, the base air temperatures, approximately 250° F, were higher than the free-stream total temperature but were still at least 300° F less than those of the closed-base configuration.

<u>0.32 Afterbody with scoops and turbine exhaust</u>. - The effects of base bleed flow on the base region temperatures of a configuration discharging propane into the base region and operating at a Mach number of 0.8 with an engine gimbal angle of 0° are presented in figure 10(a). High temperatures, of the order of 1950° F, were measured at the rake station of the closed-base turbine exhaust configuration at both 0° and 5° model angles of attack. Use of base bleed with the 0.2- and 0.5-inch-high scoops at zero angle of attack and with the 0.5-inch-high scoops at a 5° angle of attack decreased the base region temperatures to values slightly higher than free-stream total temperature (approx. 250° F). At an angle of attack of 5°, it is apparent that the 0.2-inch-high scoops did not take aboard sufficient bleed flow to prevent the recirculation and ignition of propane in the engine compartment; a temperature above 2000° F was recorded in the engine compartment.

Unfortunately, data with the closed-base, turbine exhaust configuration were not available above a Mach number of 0.8. However, the effectiveness of the scoop configuration is demonstrated at Mach numbers of 1.3, 1.6, and 2.0 (figs. 10(b) to (d)) by a comparison of the scoop configuration temperatures with those of the closed-base configuration of reference 1. At each Mach number, sufficient base bleed flow was taken aboard through either set of scoops to reduce base region temperatures from approximately 700° F to values near free-stream total temperature, approximately 125° to 175° F.

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Photographs of rocket firings for the 0.32 scoop configurations utilizing turbine exhaust and operating at a Mach number of 0.8 are presented in figure 11.

0.32 Afterbody with metered bleed flow. - The metered bleed flow was utilized with the 0.32 afterbody configuration operating with the engine gimbaled. The effects of the metered bleed flow on the engine compartment temperatures are presented in figure 12.

At Mach numbers of 2.0 and 1.4, temperatures as high as 1200° F were obtained with the 0.32 open-base configuration with zero bleed flow. These maximum temperatures were reduced substantially, to approximately 150° F, by employing bleed flow ratios of 0.025 and 0.079 at Mach numbers of 2.0 and 1.4, respectively.

At a Mach number of 0.8, temperatures were reduced from 600° F to approximately 110° F free-stream total temperature by employing a bleed flow ratio of 0.162. Although this temperature reduction was greater than that obtained with scoops, the quantity of bleed flow utilized was also significantly greater than the estimated bleed flows of the 0.5- and 0.2-inch-high scoop configurations, 0.013 and 0.005, respectively.

-0.20 Slotted afterbody. - The effects of base bleed flow on the temperatures of the -0.20 afterbody configuration with the engine at zero gimbal angle are presented in figure 13. In general, high temperatures were measured with the no-bleed configuration. Significantly lower temperatures were recorded with the slotted afterbody configuration. The measured temperatures of both configurations, however, were affected by the heat sink characteristics of the cast-iron insert around the base. At a Mach number of 2.0, for example, temperatures of the no-bleed configuration, measured 1/8 inch downstream of the base, were as high as 1640° F, while air temperatures measured at the base surface were only as high as 840° F. Likewise, the air temperatures 1/8 inch downstream of the divergent portion of the cast-iron ring were as high as 2150° F, while those at the surface of the ring were only as high as 640° F. At this Mach number, the use of bleed flow reduced the air temperatures at the base as well as those on the divergent portion of the cast-iron ring to approximately 250° F.

Temperatures of the slotted afterbody measured at the rake station 1.42 inches upstream of the base were of the order of free-stream total temperature, approximately 150° F. Although the rake of the slotted afterbody was not located near the divergent portion of the cast-iron support, the measured rake temperatures should be indicative of the magnitude of the air temperatures that would be measured by a thermocouple located close to the surface of the divergent portion of the cast-iron sections. Photographs showing the effect of bleed flow on base burning at a Mach number of 2.0 are presented in figure 14.

Pressures

Rocket-engine external pressure distributions with 0.32 afterbody configurations. - The effects of base bleed flow on the rocket-engine pressure distributions when the engine is at zero gimbal angle are presented in figure 15. Uniform pressure distributions were obtained with the no-bleed configuration of reference 1. However, the use of base bleed flow through afterbody slots or the 0.5-inch-high scoops resulted in appreciable pressure disturbances over the engine surface; only small disturbances were induced by the use of the 0.2-inch-high scoops. Metered bleed flow introduced no pressure disturbances (data not presented). Thus, the manner in which bleed was ducted through the engine compartment significantly affected the engine pressure distributions.

Rocket-engine external pressure distributions of 0.58 afterbody configuration. - Typical effects of metered bleed flow on the pressure distributions of the 0.58 afterbody configuration at zero angle of attack and with the engine gimbaled to 4° are presented in figure 16. With zero bleed flow, large pressure disturbances were obtained near the exit plane of the engine at supersonic speeds. Use of bleed flow reduced these disturbances and also generally reduced the pressure difference between the windward and leeward side of the engine.

Aerodynamic moments, obtained by the integration of the external pressure forces, are plotted in figure 17 as moment coefficients against bleed flow ratios. At all Mach numbers and for both 0° and 5° angles of attack, bleed flow was effective in reducing the magnitude of the moment coefficients introduced by gimbaling. This was true for the generally observed positive coefficient as well as for the negative values at an angle of attack of 5° and a Mach number of 2.0. Extrapolation of the curves would indicate that bleed flows substantially larger than those tested could result in moments of greater magnitude (and opposite in direction) than the no-bleed moments.

Base pressures. - Although base pressures do not appreciably affect the performance of missiles, the base pressure is part of the environment of the exhaust jet and as such is of interest. Base pressure variations with bleed flow ratios of the afterbody configurations tested are therefore presented in figure 18. Base pressure ratios p_b/p_0 for the 0.32 and -0.20 slotted afterbodies are not shown in figure 18. At a Mach number of 2.0, base pressure ratios of 0.675 and 0.742 were obtained with the 0.32 and -0.20 slotted afterbodies, respectively. At a Mach number of 0.8, a base pressure ratio of approximately 0.95 was obtained with both configurations.

SUMMARY OF RESULTS

A study was made of the effects of ducting bleed flow into the base region of several simulated missile configurations utilizing a liquid-oxygen - JP-4 fuel rocket engine. Nominal afterbody diameter was approximately 8 inches. Bleed air was obtained from either a metered source external to the tunnel or through afterbody slots and scoops. Base region temperature and pressure data obtained through a Mach number range of 0.8 to 2.0 indicated the following:

1. Bleed air was effective in reducing base region temperatures of the configurations tested. For example, the temperatures of the configuration having an engine retraction of 0.20 caliber upstream of the afterbody base were reduced from values as high as 2000° F to near free-stream total temperature. Likewise, temperatures of a configuration having an engine projection of 0.32 caliber downstream of the afterbody base were reduced from values as high as 700° F to approximately free-stream total temperature.

2. At a Mach number of 0.8, with a configuration discharging simulated turbine-drive gases into the base region, bleed flow through the 0.5-inch-high scoop configuration prevented the recirculation and burning of these combustible gases in the base region, thereby reducing base region temperatures from 1950° to 250° F.

3. With the 0.32 afterbody configuration, the axial pressure distributions over the outside of the engine were dependent upon the manner in which the bleed air was ducted through the engine compartment. At zero angle of attack and zero engine gimbal angle, ducting bleed flow through afterbody slots or the 0.5-inch-high scoops and then through the engine compartment resulted in large axial pressure gradients along the engine surface. Reducing the scoop height to 0.2 inch resulted in a more uniform engine pressure distribution. Metered bleed flow, however, introduced no pressure disturbances.

4. With the 0.58 afterbody configuration, which did not experience base heating problems, bleed flow from the metered source was effective in reducing engine aerodynamic moments caused by the free-stream flow.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, December 4, 1959

REFERENCES

1. Chiccine, Bruce G., Valerino, Alfred S., and Shinn, Arthur M.: An Experimental Investigation of Base Heating and Rocket Hinge Moments for a Simulated Missile Through a Mach Number Range of 0.8 to 2.0. NASA TM X-82, 1959.

2. Simon, Paul C., and Kowalski, Kenneth L.: Charts of Boundary-Layer Mass Flow and Momentum for Inlet Performance Analysis - Mach Number Range 0.2 to 5.0. NACA TN 3583, 1955.

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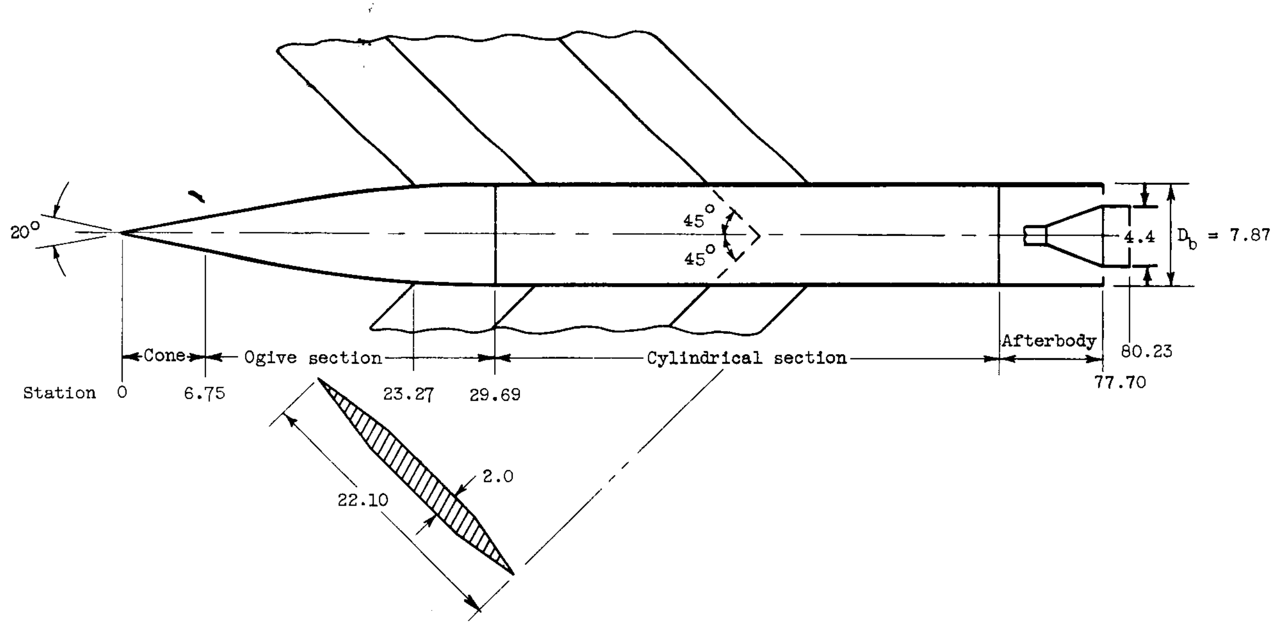
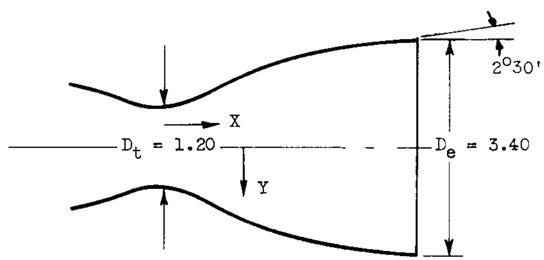


(a) Assembly.



X	Y
0	0.600
.090	.617
.275	.691
.609	.826
.944	.960
1.322	1.109
1.699	1.233
1.968	1.328
2.408	1.450
2.756	1.530
3.048	1.591
3.343	1.639
3.581	1.665
3.819	1.683
4.057	1.698
4.132	1.700

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(b) Engine nozzle internal contour.

Figure 1. - Schematic diagram of model. (Dimensions in inches except where noted.)

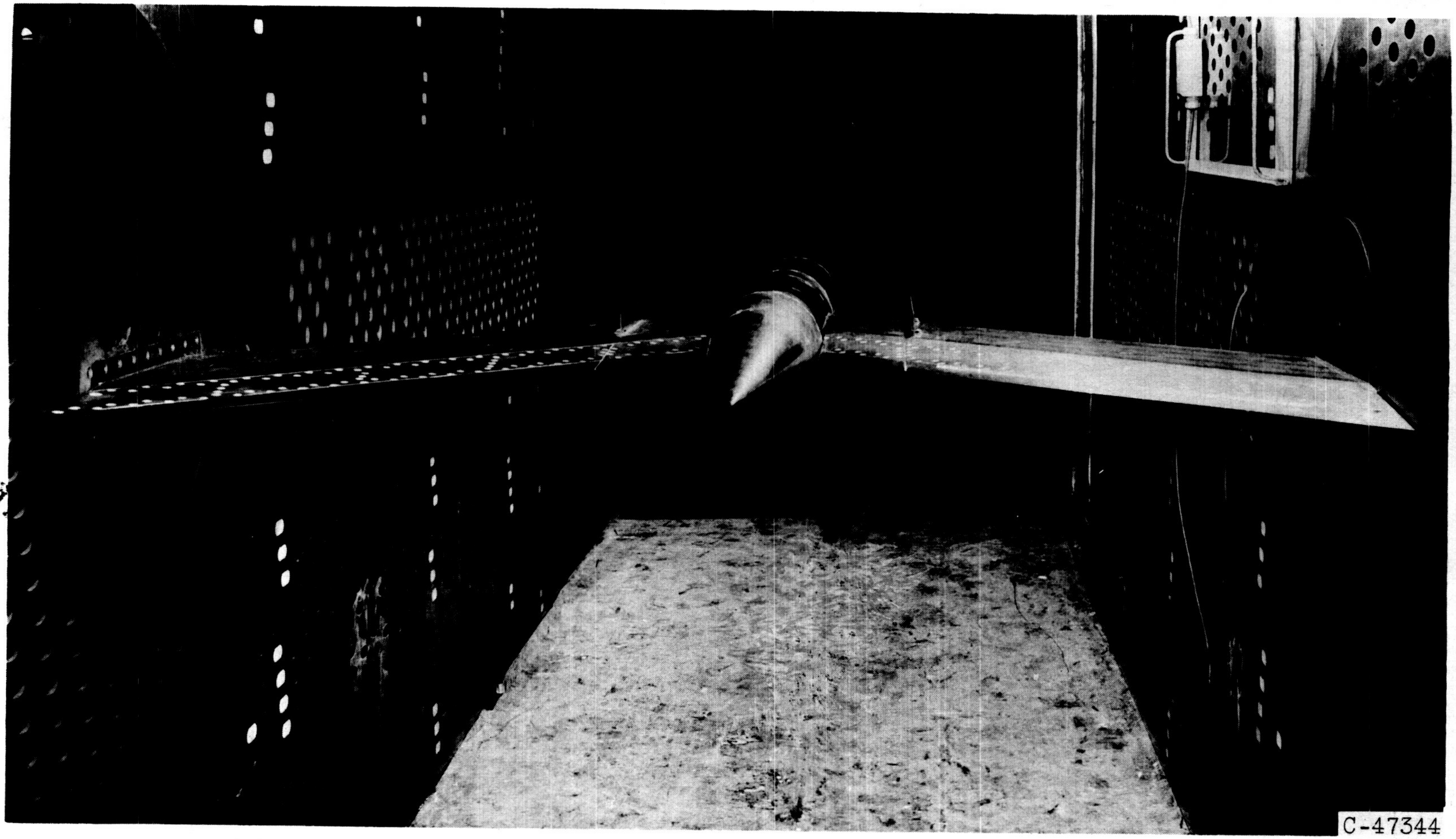

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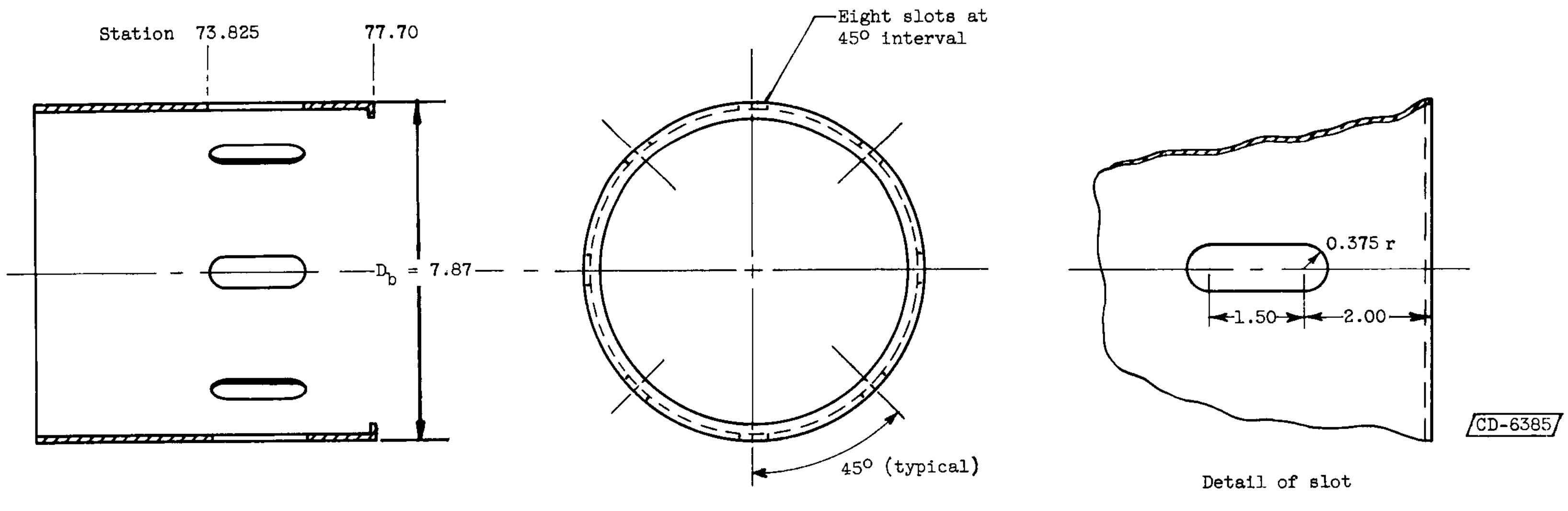
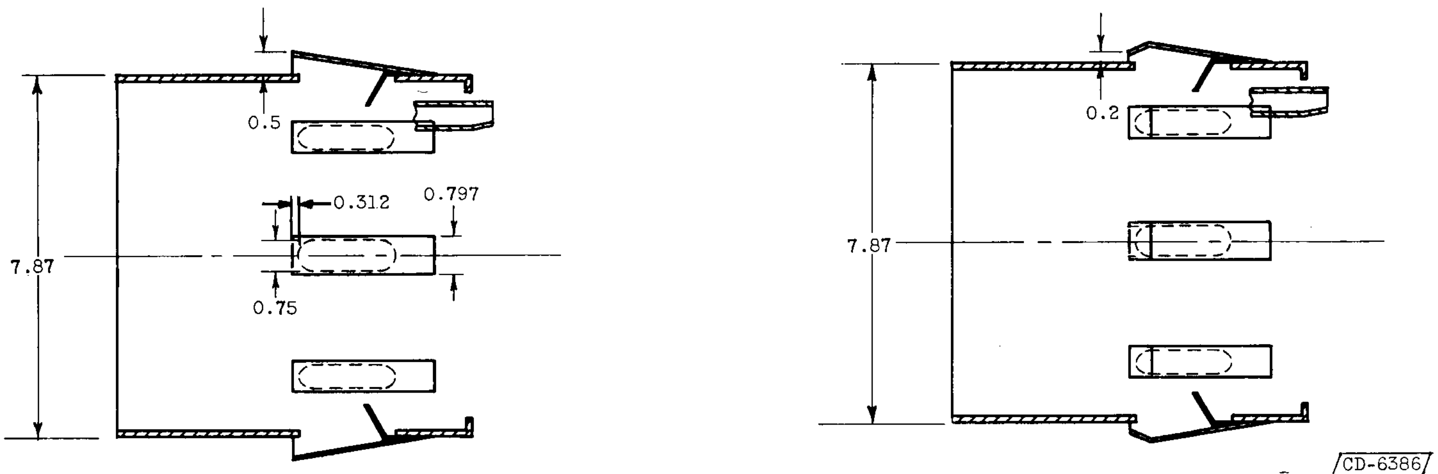
Figure 2. - Model assembly in test section.



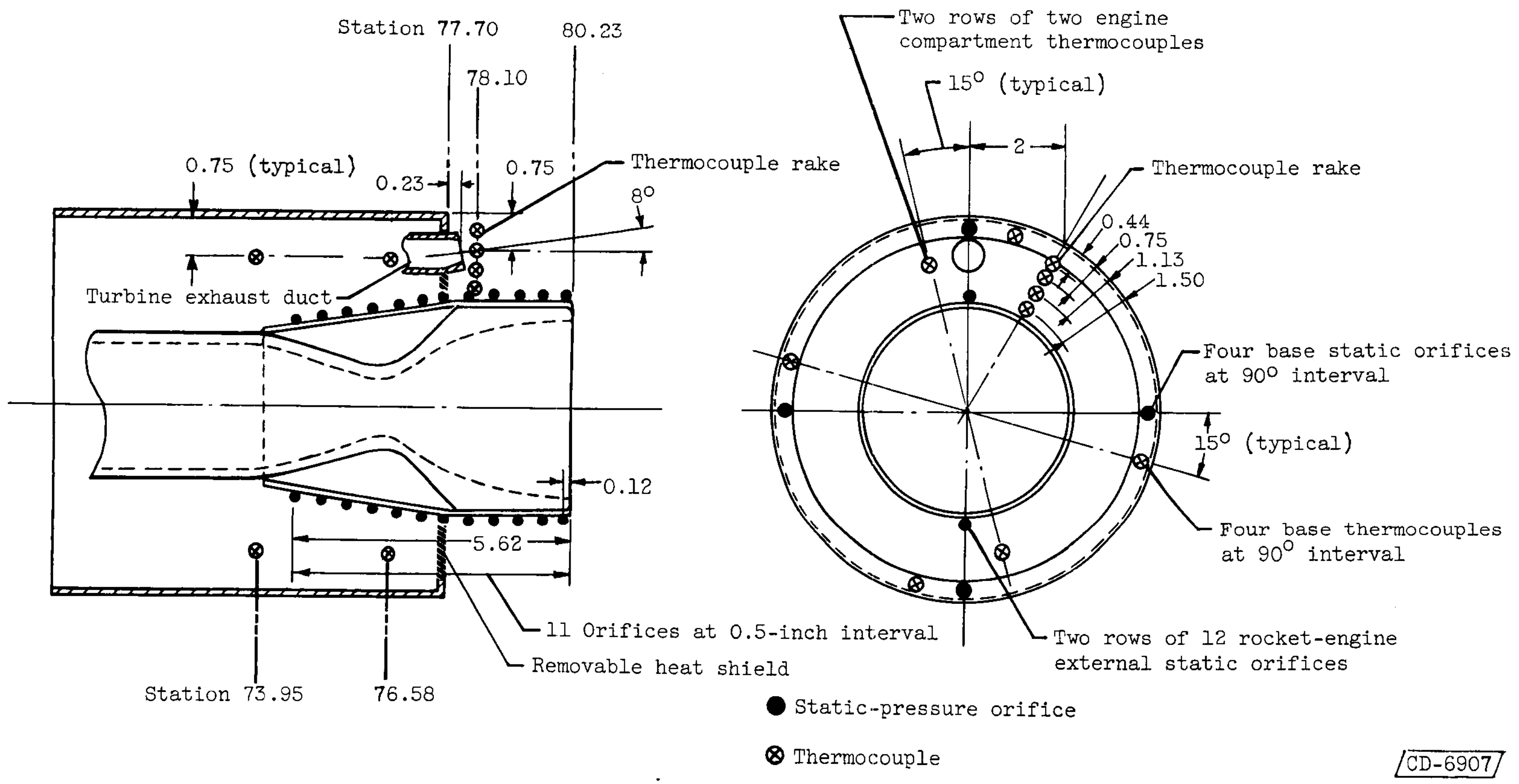
(a) Slotted afterbody.



(b) 0.5-Inch-high scoop configuration.

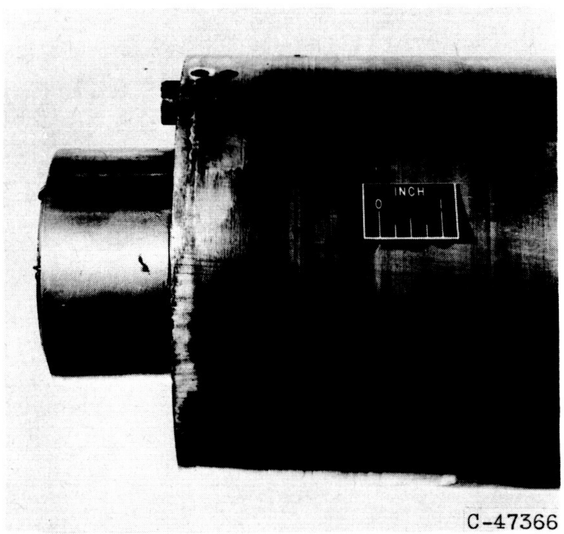
(c) 0.2-Inch-high scoop configuration.

Figure 3. - 0.32 Afterbody configurations. (Dimensions in inches except where noted.)

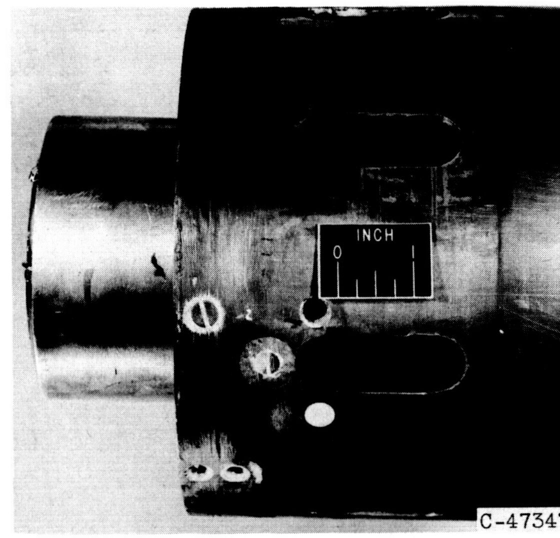


(d) Closed-base turbine exhaust configuration showing typical static-pressure and thermocouple instrumentation for all 0.32 afterbody configurations.

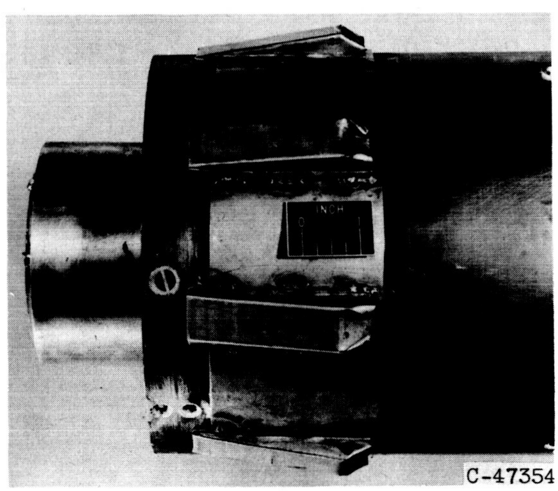
Figure 3. - Concluded. 0.32 Afterbody configurations. (Dimensions in inches except where noted.)



(a) Closed-base turbine exhaust configuration.



(b) Slotted afterbody.



(c) Afterbody with 0.2-inch-high scoops.

Figure 4. - Photographs of 0.32 afterbody configurations.

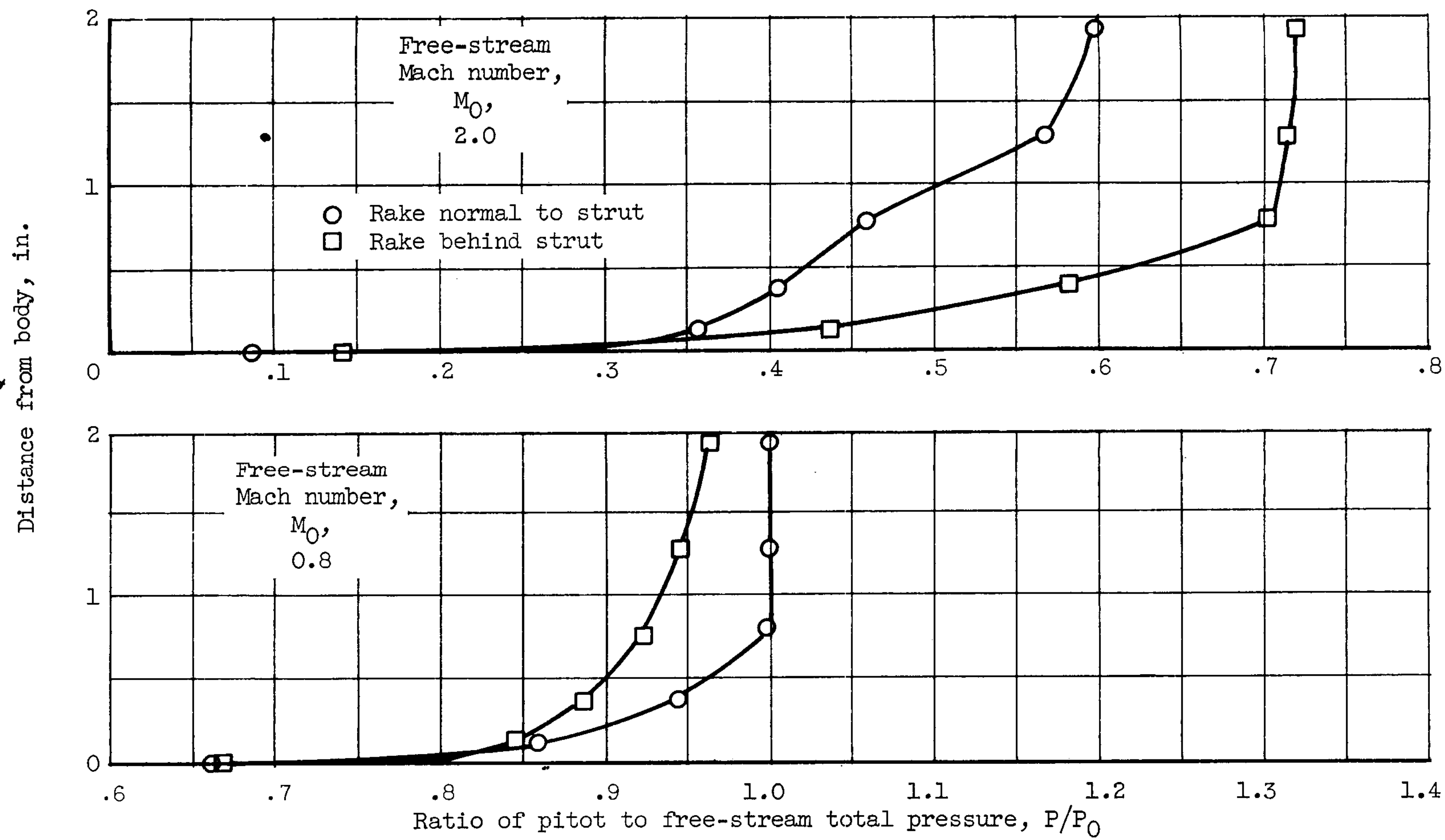


Figure 5. - Typical pitot pressure profiles of model boundary layer at model station 59.

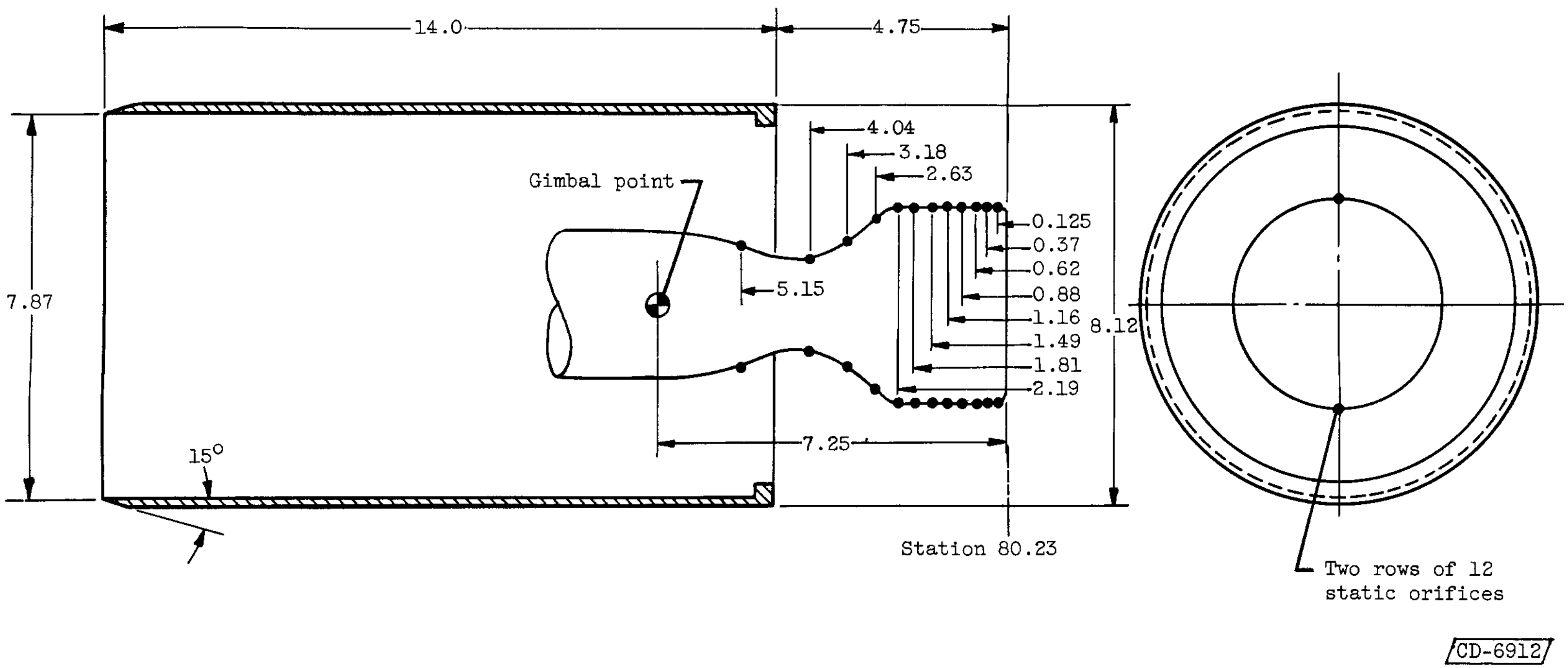
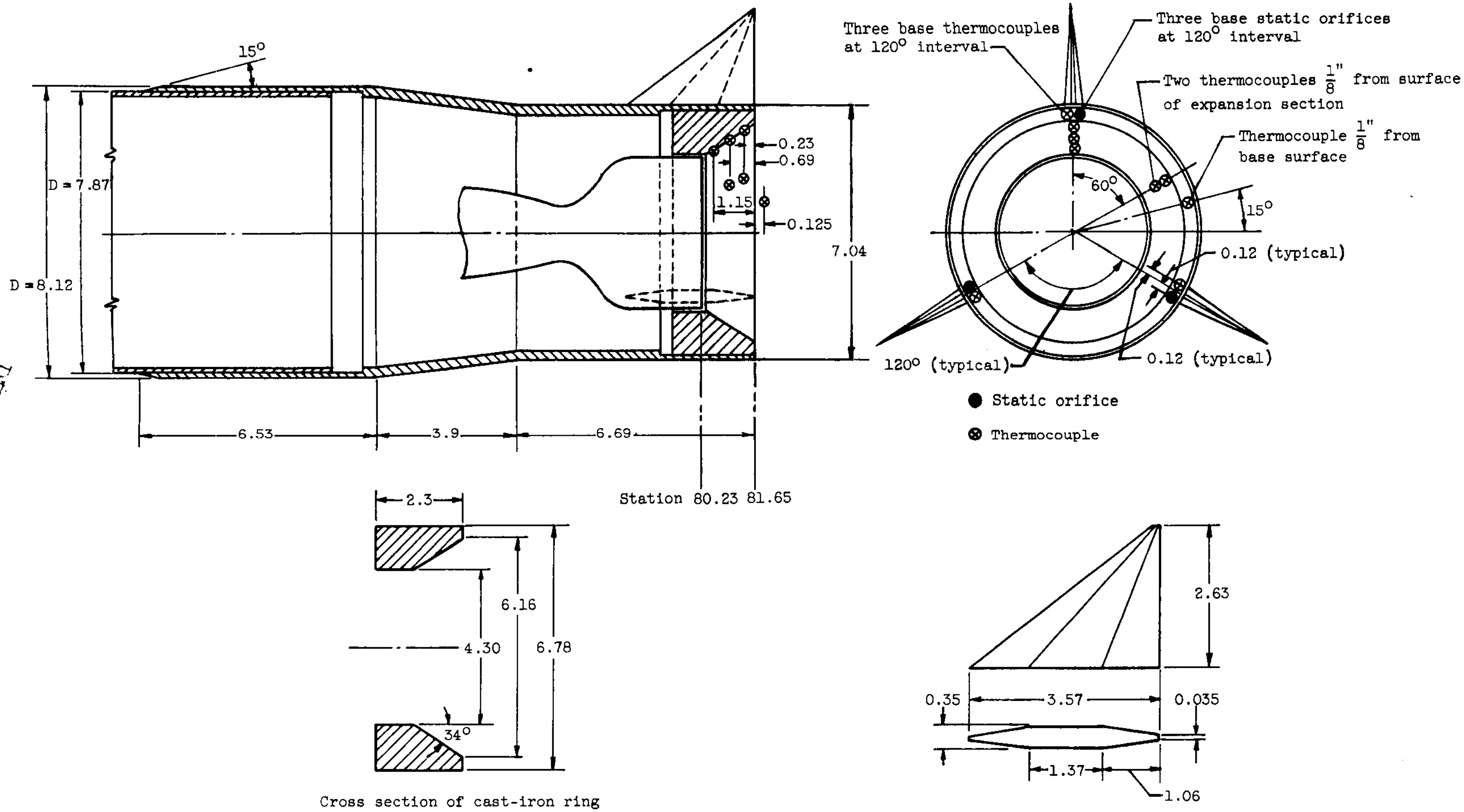
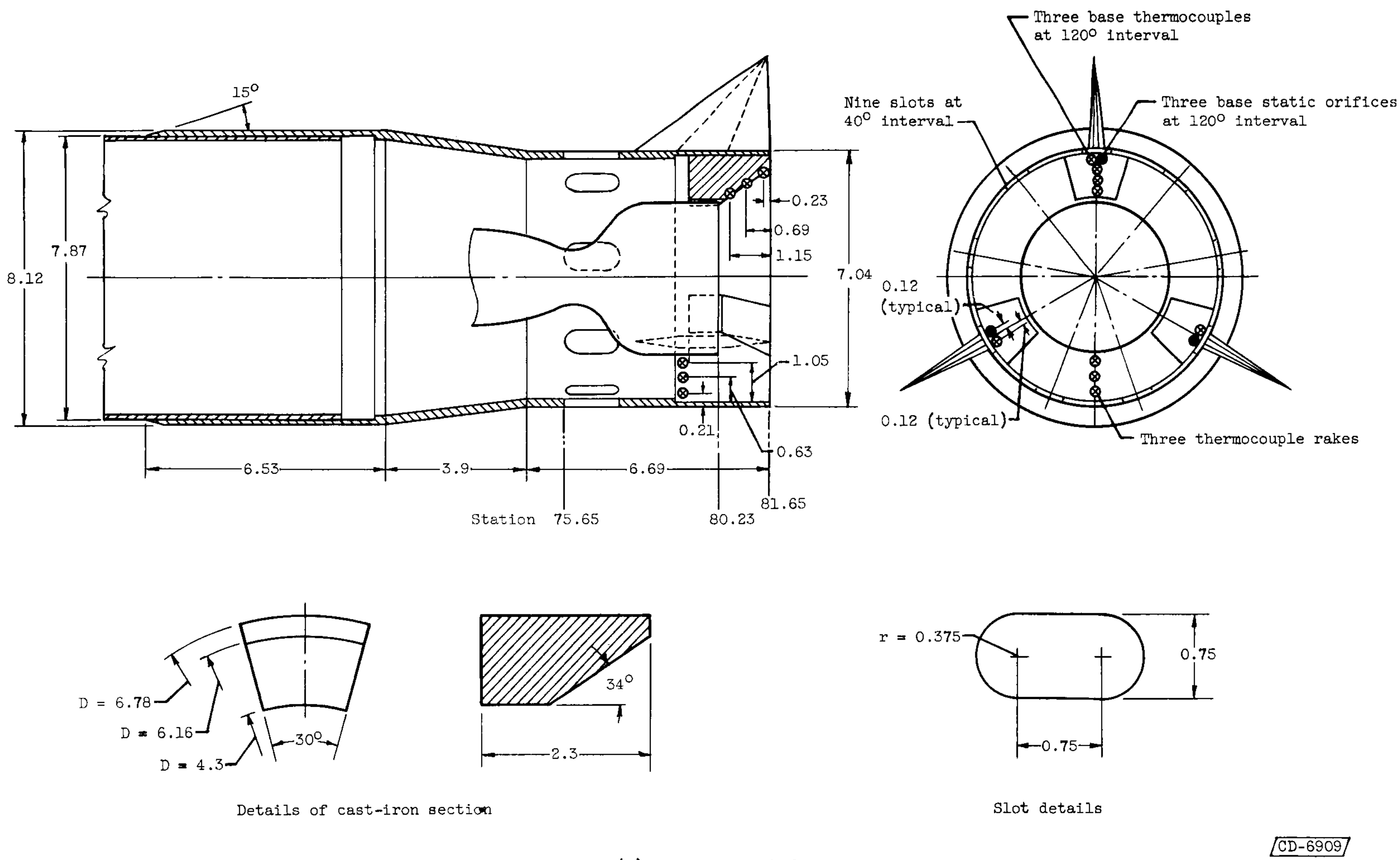


Figure 6. - 0.58 Afterbody configuration. (Dimensions in inches except where noted.)



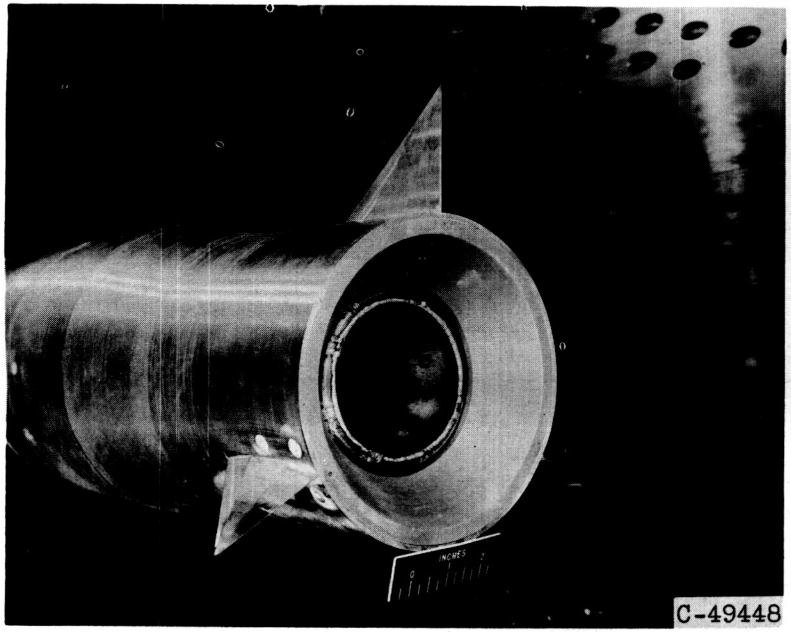
(a) No-bleed configuration.

Figure 7. - -0.20 Afterbody configuration. (Dimensions in inches except where noted.)

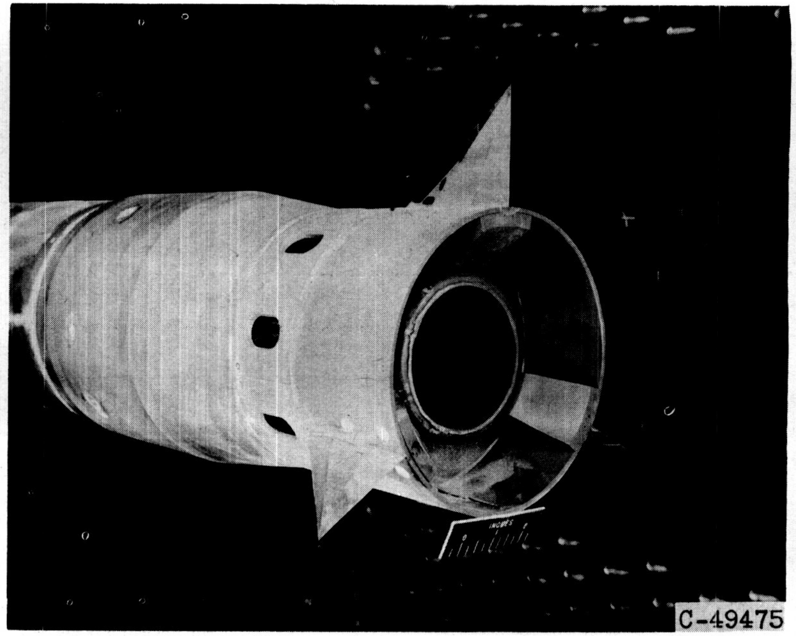


(b) Slotted afterbody.

Figure 7. - Concluded. -0.20 Afterbody configuration. (Dimensions in inches except where noted.)

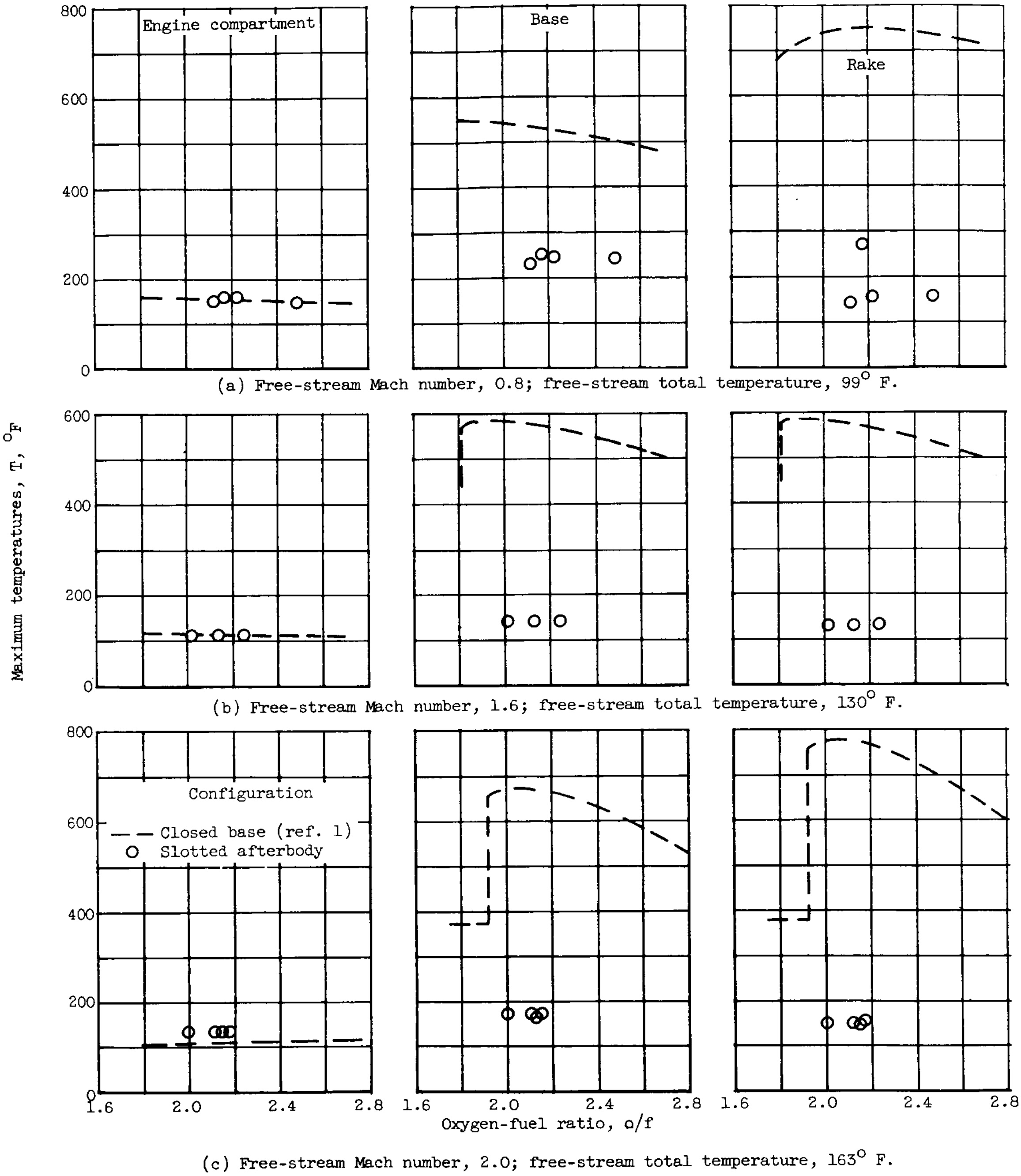


(a) -0.20 Afterbody.



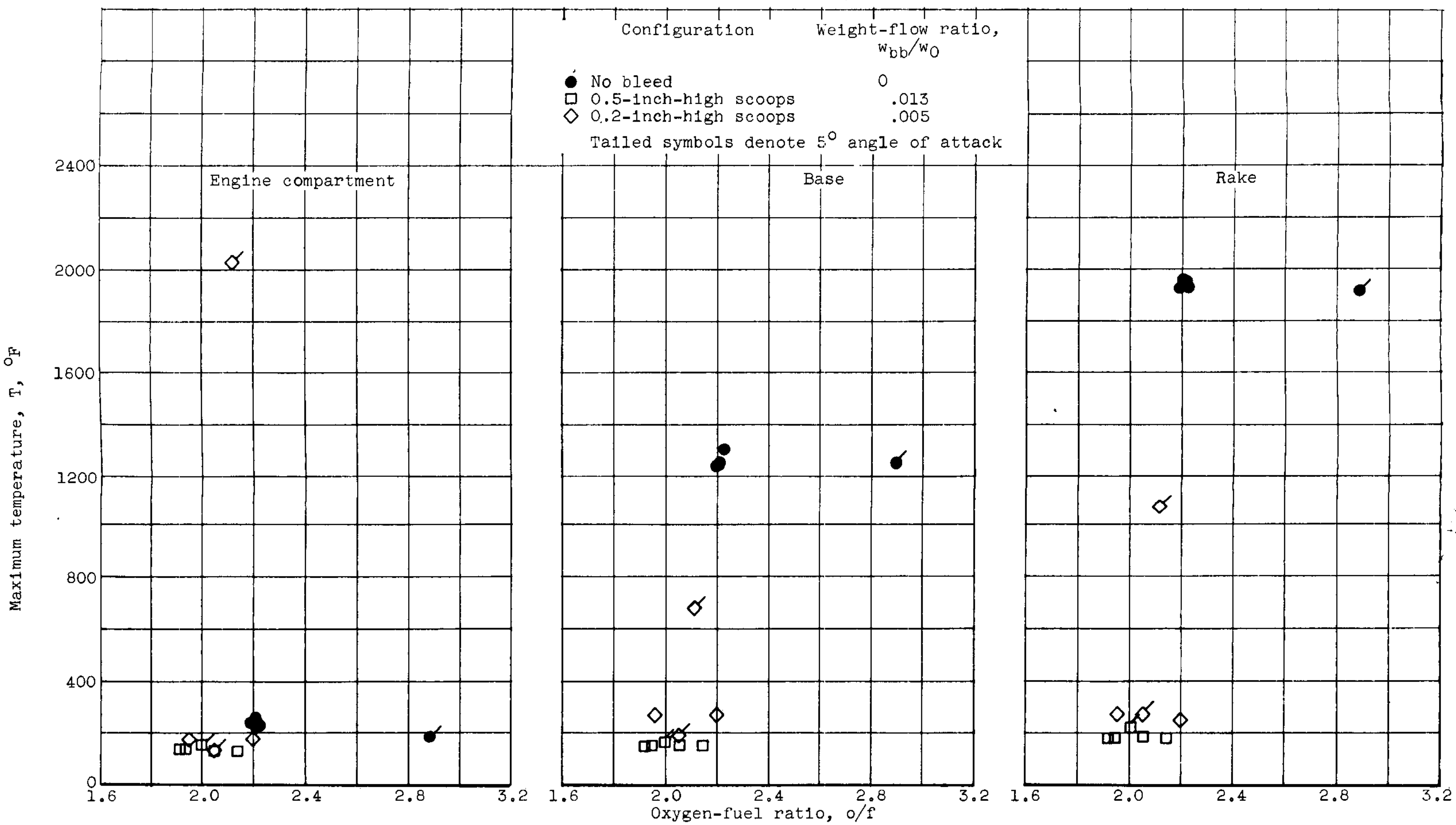
(b) -0.20 Slotted afterbody.

Figure 8. - Photographs of -0.20 afterbody configurations.



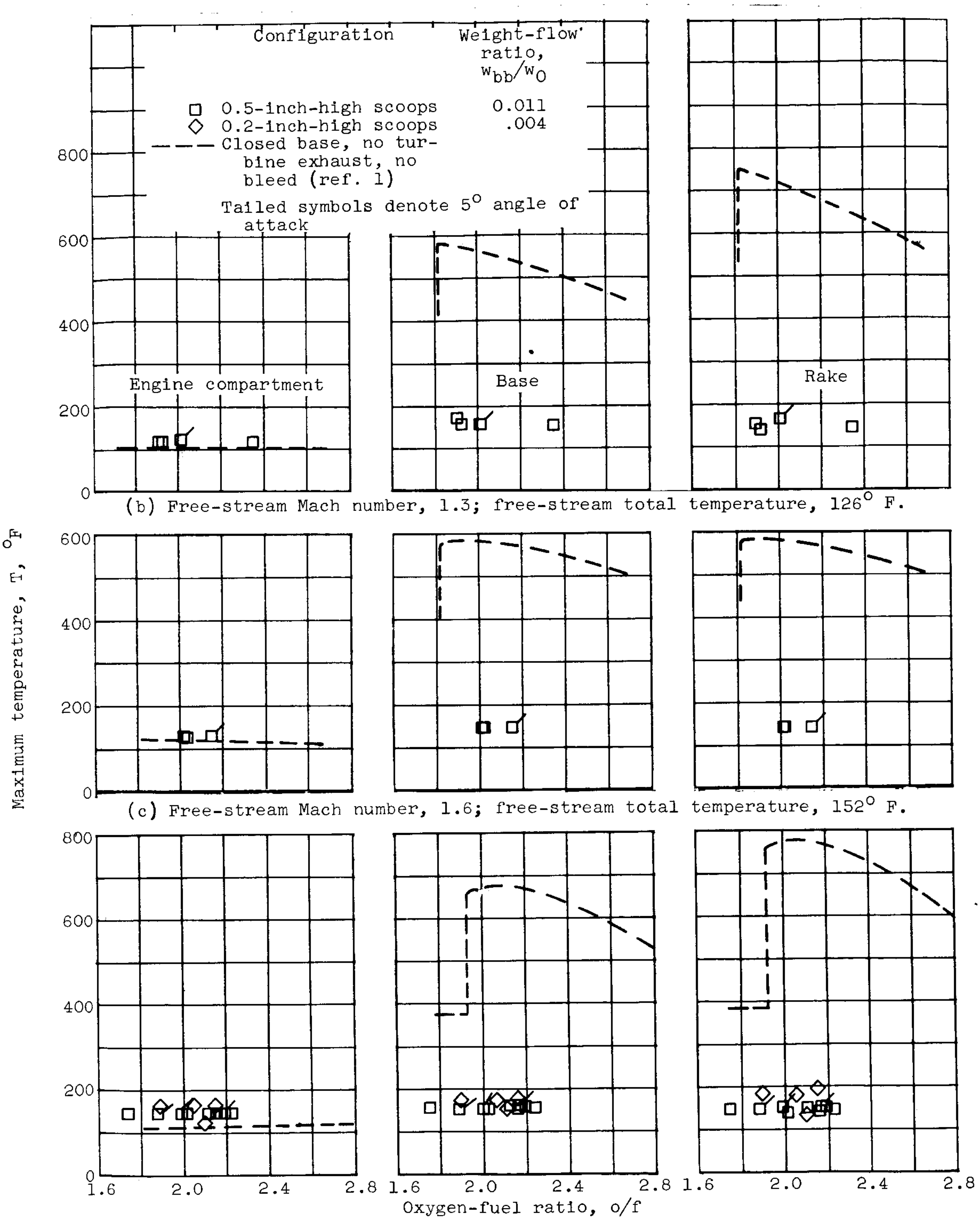
(c) Free-stream Mach number, 2.0; free-stream total temperature, 163° F.

Figure 9. - Effects of bleed flow through slots on temperatures of 0.32 afterbody configuration operating at zero angle of attack with zero engine gimbal angle.



(a) Free-stream Mach number, 0.8; free-stream total temperature, 110° F.

Figure 10. - Effects of bleed flow through scoops on temperatures of 0.32 afterbody configuration discharging propane into base region with engine at zero gimbal angle.

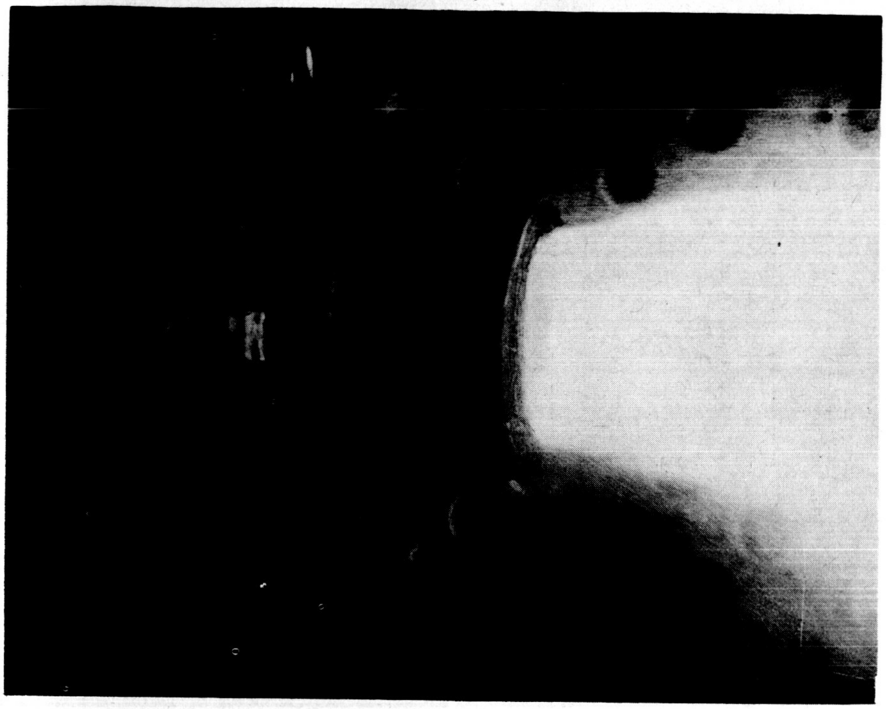


(b) Free-stream Mach number, 1.3; free-stream total temperature, 126° F.

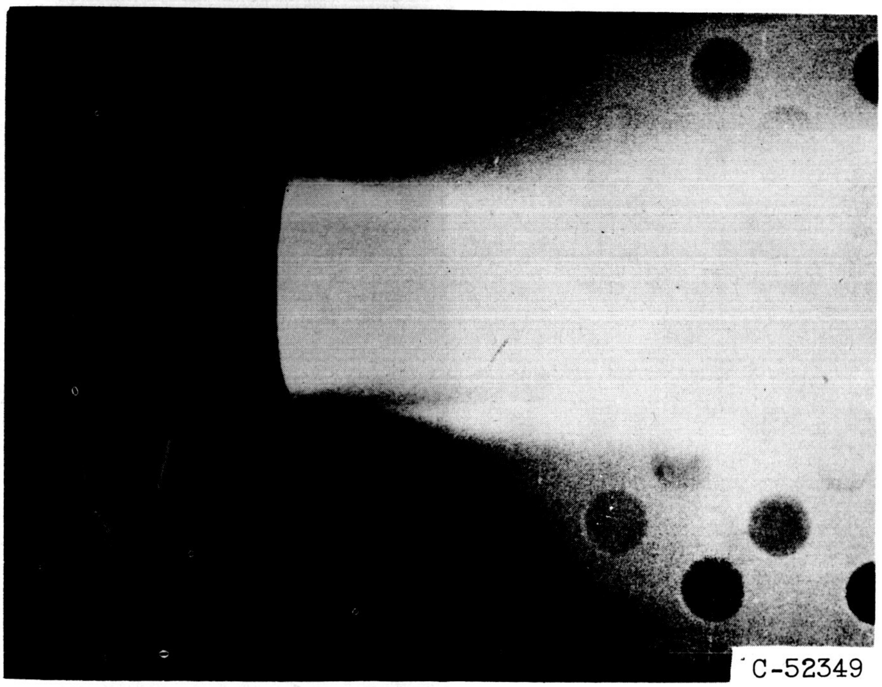
(c) Free-stream Mach number, 1.6; free-stream total temperature, 152° F.

(d) Free-stream Mach number, 2.0; free-stream total temperature, 174° F.

Figure 10. - Concluded. Effects of bleed flow through scoops on temperatures of 0.32 afterbody configuration discharging propane into base region with engine at zero gimbal angle.



0.2-Inch-high scoops; oxygen-fuel ratio, 2.20



0.5-Inch-high scoops; oxygen-fuel ratio, 1.91

Figure 11. - 0.32 Afterbody scoop configurations utilizing turbine exhaust and operating at free-stream Mach number of 0.8 and zero angle of attack.

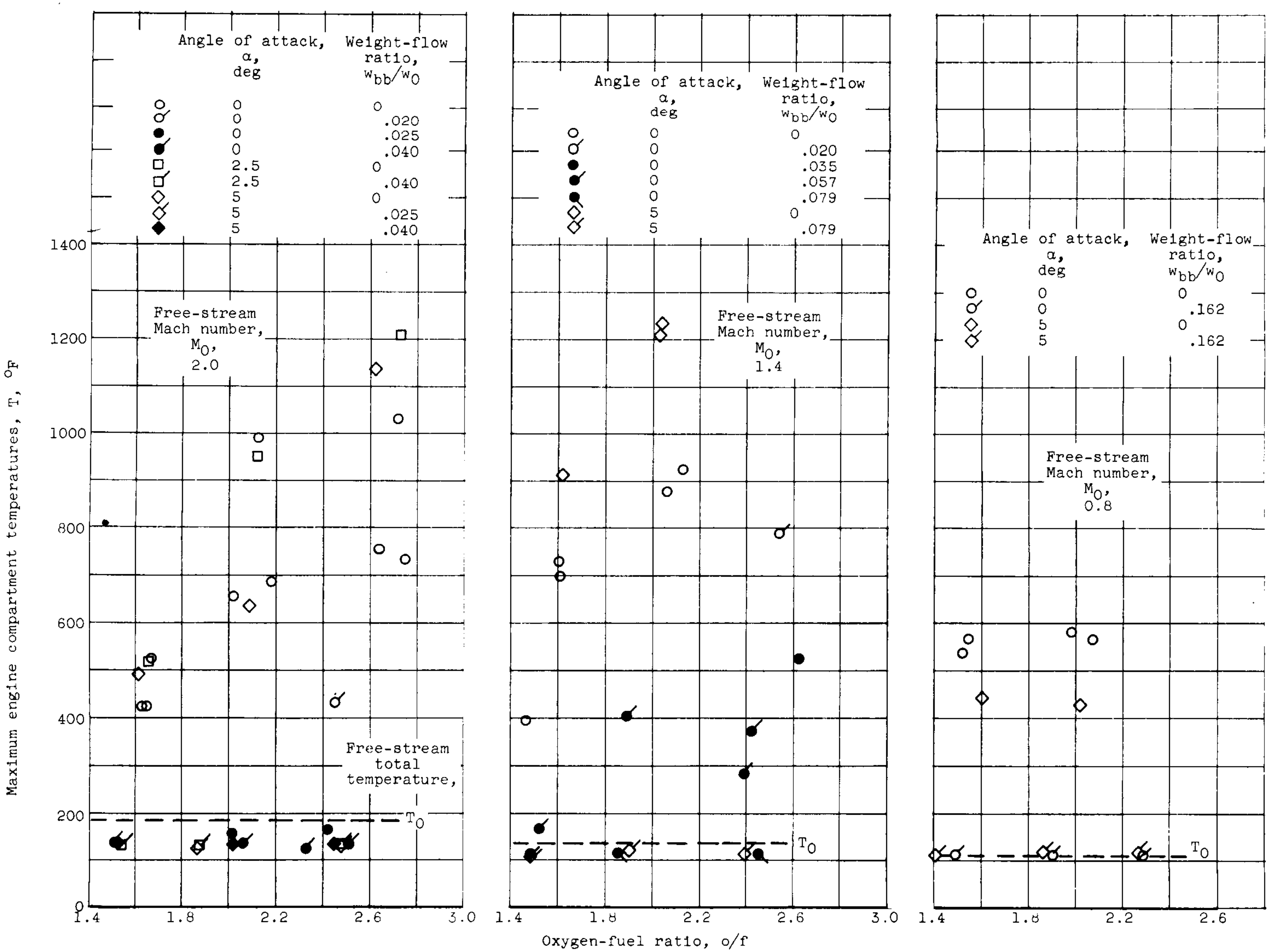
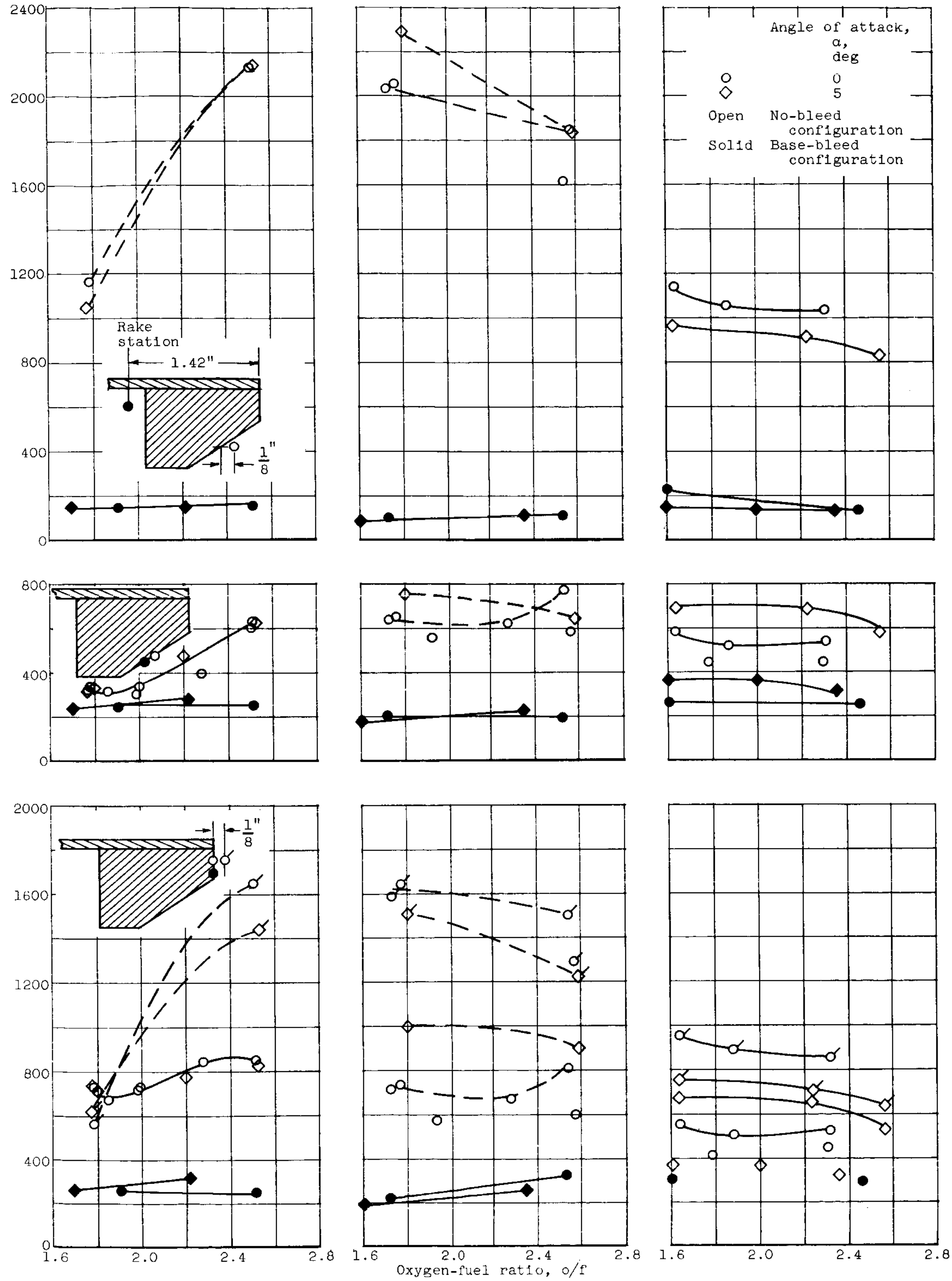


Figure 12. - Effects of metered bleed flow on engine compartment temperatures of 0.32 afterbody configuration with engine at 4° gimbal angle.

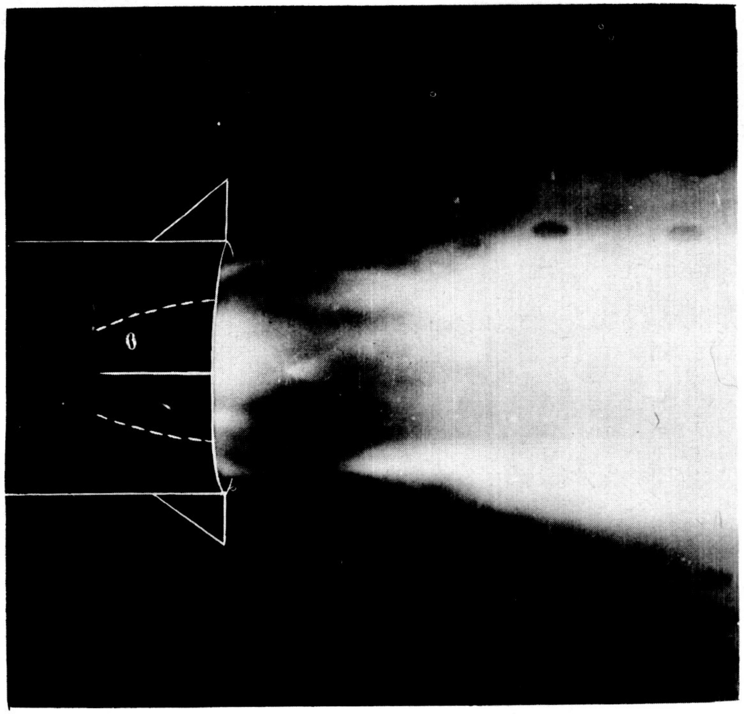


(a) Free-stream Mach number, 2.0; free-stream total temperature, 150° F.

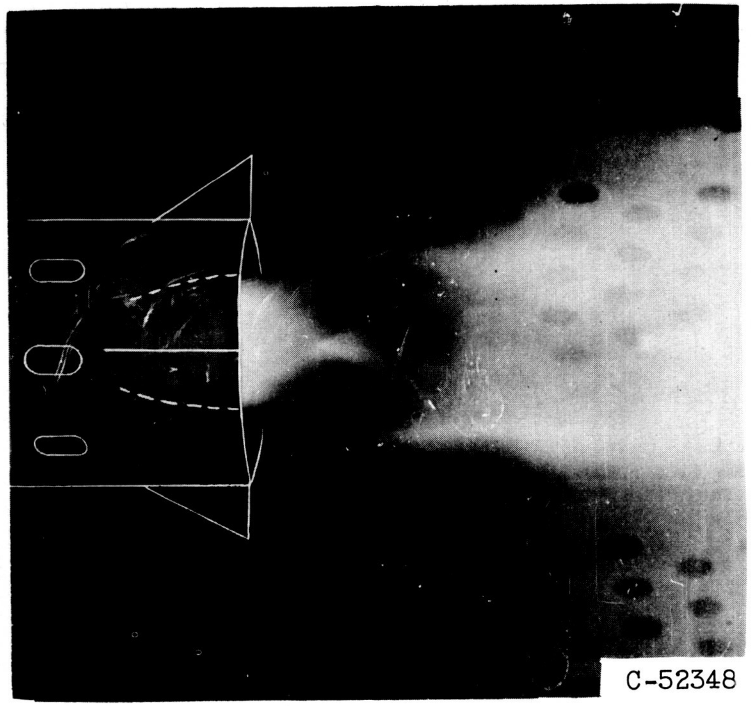
(b) Free-stream Mach number, 1.4; free-stream total temperature, 100° F.

(c) Free-stream Mach number, 0.8; free-stream total temperature, 90° F.

Figure 13. - Effects of bleed flow through slots on temperatures of -0.20 afterbody configuration at zero angle of attack with zero engine gimbal angle.

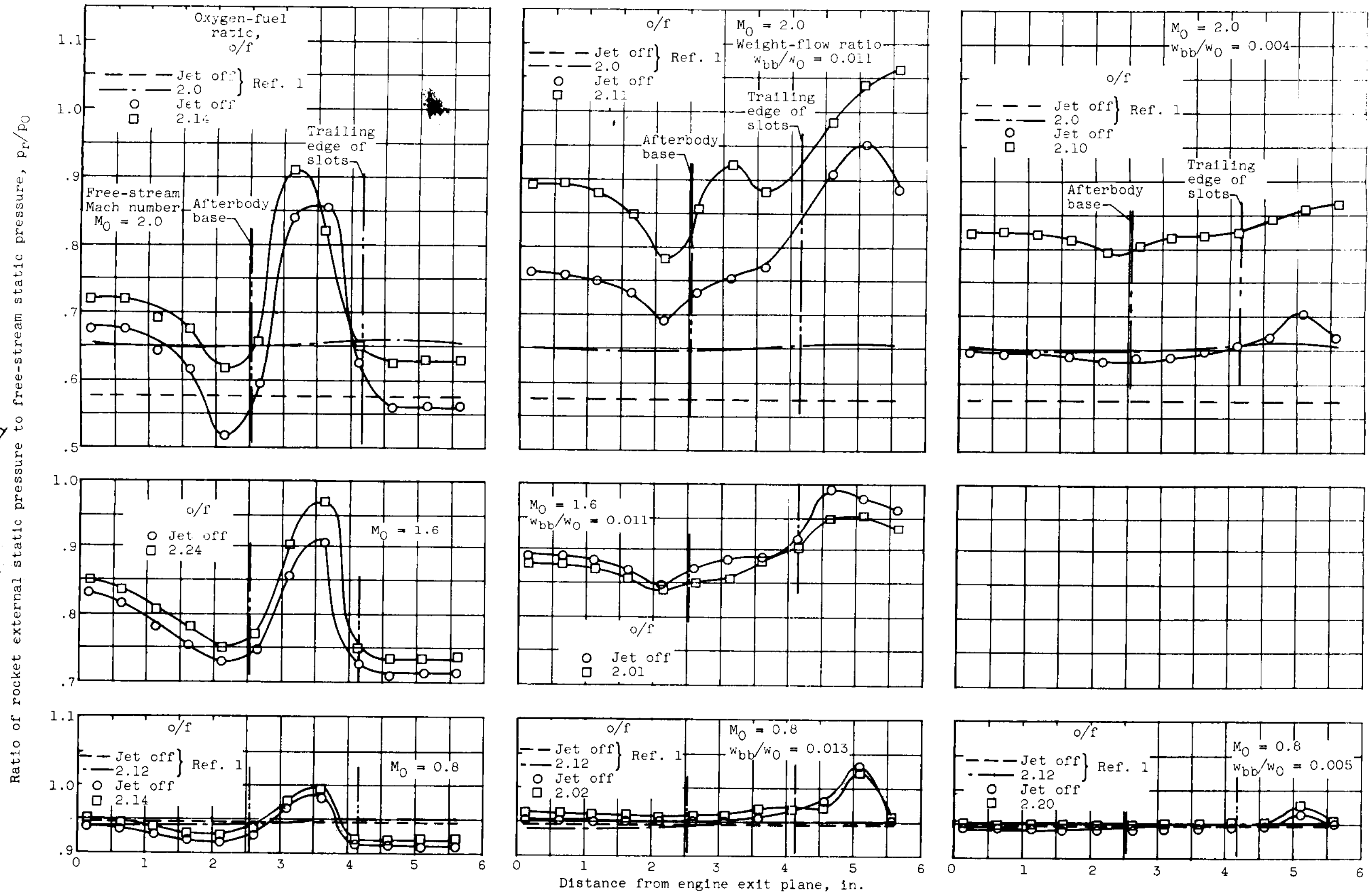


No-base-bleed configuration; base burning



Slotted afterbody configuration; no base burning

Figure 14. - -0.20 Afterbody configurations operating at Mach number of 2.0, oxygen-fuel ratio of 2.51, and zero angle of attack.

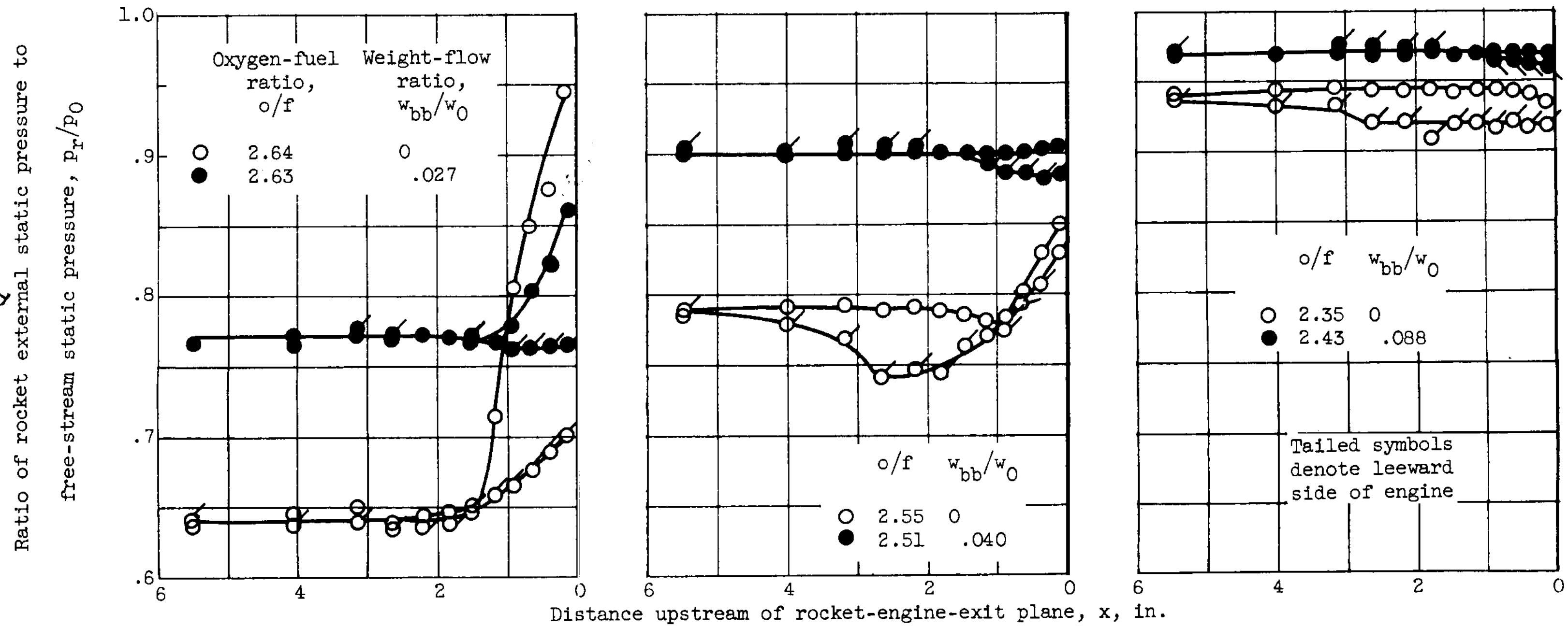


(a) Slotted afterbody.

(b) Afterbody with 0.5-inch-high scoops.

(c) Afterbody with 0.2-inch-high scoops.

Figure 15. - Effects of bleed flow through slots or scoops on engine external pressure distributions of 0.32 afterbody configuration. Zero angle of attack; zero engine gimbal angle.



(a) Free-stream Mach number, 2.0. (b) Free-stream Mach number, 1.4. (c) Free-stream Mach number, 0.8.

Figure 16. - Effects of metered bleed flow on engine external pressure distributions of 0.58 afterbody configuration operating at zero angle of attack with engine at 4° gimbal angle.

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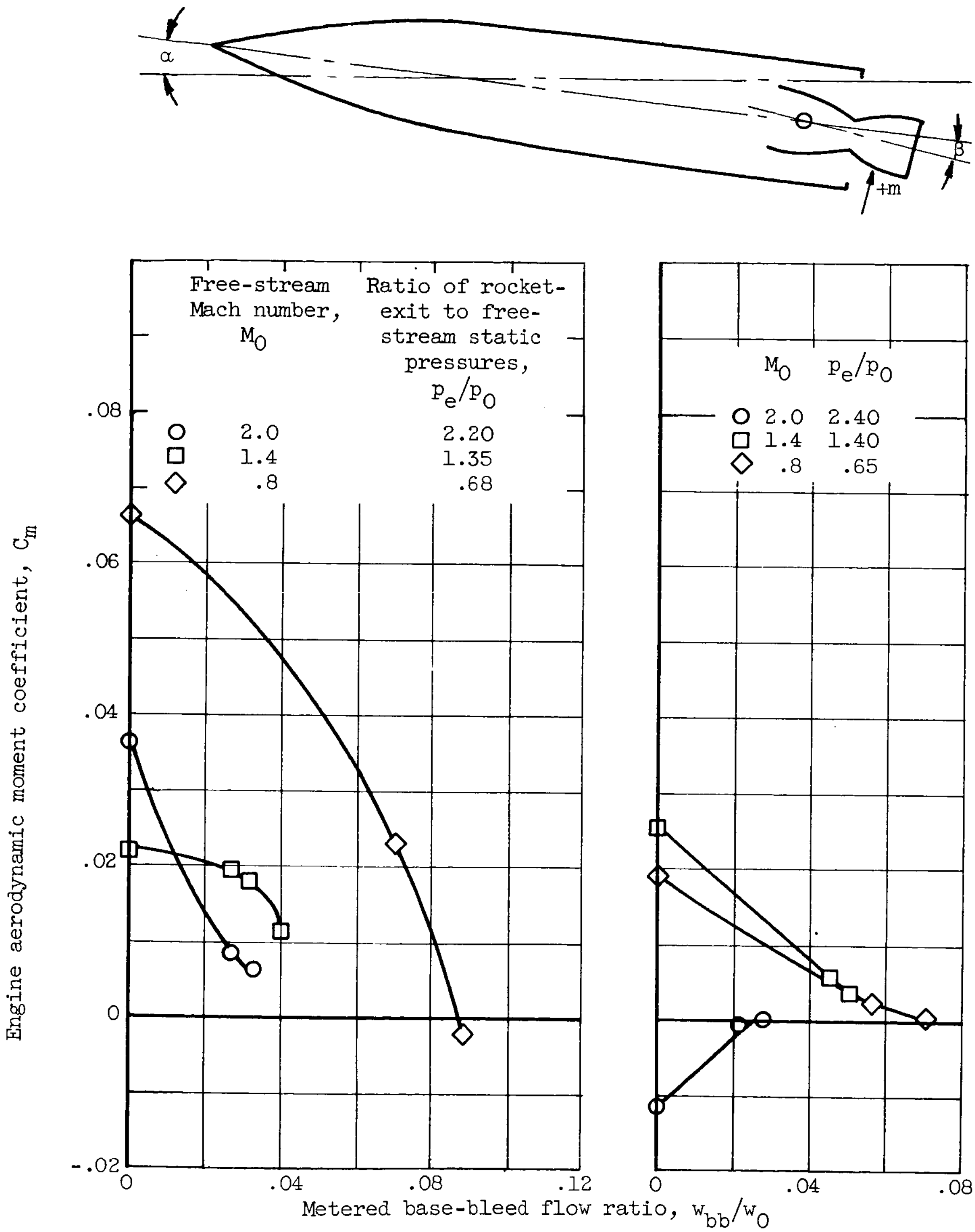
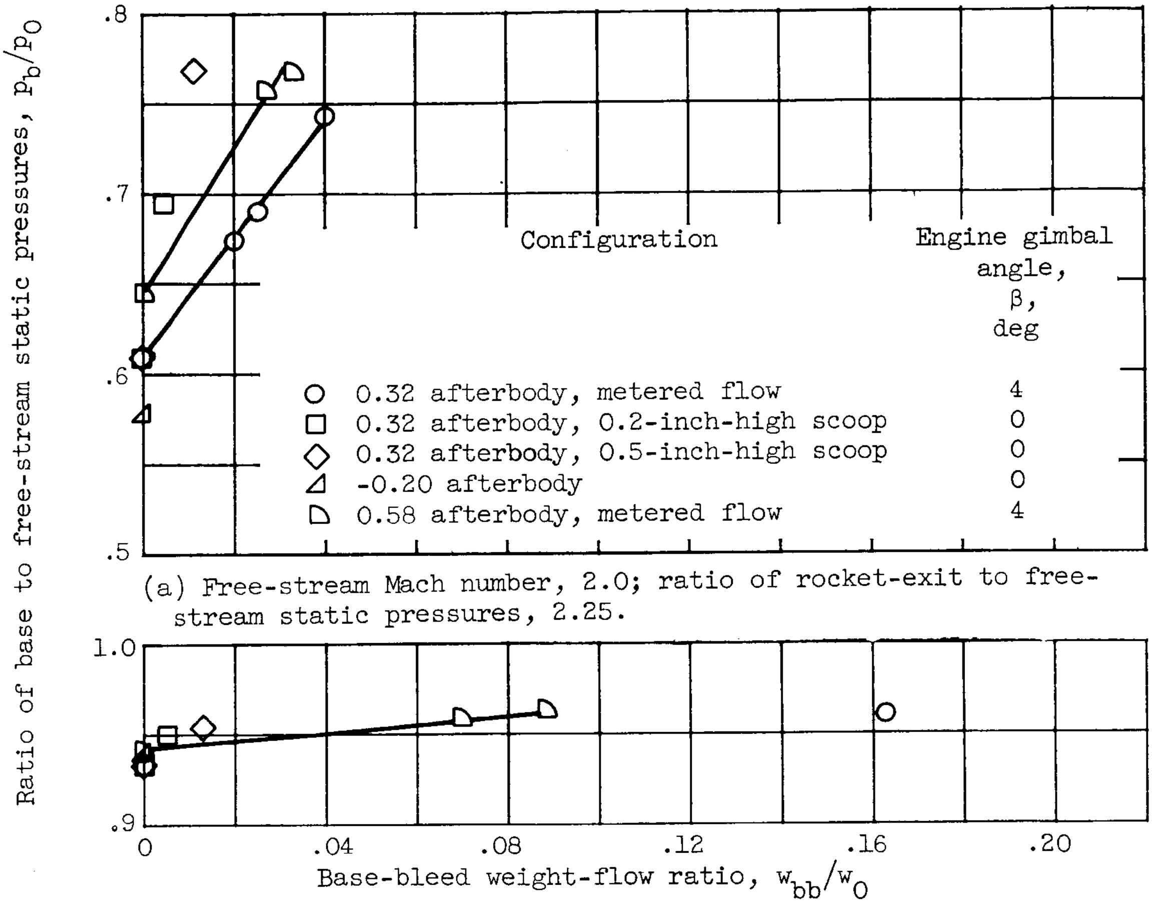


(a) Angle of attack, 0°.

(b) Angle of attack, 5°.

Figure 17. - Effects of metered bleed flow on engine aerodynamic moment coefficients of 0.58 afterbody configuration with engine at 4° gimbal angle.



(a) Free-stream Mach number, 2.0; ratio of rocket-exit to free-stream static pressures, 2.25.

(b) Free-stream Mach number, 0.8; ratio of rocket-exit to free-stream static pressures, 0.6.

Figure 18. - Effect of base bleed on base pressure. Zero angle of attack.